New Millennium Program

New Millennium Program (NMP)

Space Environments on Electronic Components Guidelines

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Space Environments on Electronic Components Guidelines, Version 0

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Introduction

The New Millennium Program has adopted the *"The Effects of Space Environments on Electronic Components"* document for project managers and system engineers to use as a starting point and *guidelines* for addressing space environments vs stresses on electronics factors and impacts associated with developing their space validation experiments.

The Effects of Space Environments on Electronic Components

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Disclaimer

This document is intended to provide a general introduction to the space environments and their influence on the performance of electronic technologies. Qualification of specific technologies requires an accurate description of mission requirements and testing of the specific electronic components relative to the mission requirements.

CONTENT

I. INTRODUCTION

One of the challenges in space qualification is to define the operational environment of a part such that it is tested to the limits of a mission without requiring expensive overdesign. To aid this, this document defines, discusses and recommends environmental design and verification requirements for using microelectronic components and assemblies in space environments.

The characteristics of a variety of environments, relevant to both Earth and space scientific missions, are enveloped within typical representative parameters. Discussed are: radiation, thermal, vibration, electrostatic, electromagnetic, and electrical environments. Presented are the typical performances of devices classified by technology for each of the relevant conditions. Distinction is made between monolithic devices and assemblies, as they require emphasis on different environments.

Disclaimer:

This document is built on a foundation of experimental data from past and current JPL missions, the present knowledge and understanding of space environments, and theoretical models with predictive capability used in mission planning. The purpose of this document is to assist the understanding of the typical environments by presenting realistic and typical guidelines. For better understanding of space environments it is recommended that the reader refers to sources such as "Spacecraft-Environment Interactions" by D. Hastings and H. Garrett (Cambridge Univ. Press, NY 1996).

It should be stressed that the environmental characteristics may vary greatly within the same category, as defined for the purpose of this document, and the data are by no means intended as qualification guidelines. The actual application of each microelectronic component, sub-system or system can influence the environments, to which it is exposed, and therefore may change the test requirements.

II. RADIATION ENVIRONMENT

1. Definition of radiation environments

The sources for radiation in interplanetary space are the Galactic background radiation and the Sun. The Galactic background radiation is practically uniform in the range of all planetary orbits. The Sun is an active source of radiation and plasma, emitting protons, electrons, photons, and heavy ions. The solar radiation intensity decreases with increasing distance from the Sun. The characteristics of these two sources determine the radiation environment in interplanetary space. However, this is insufficient to give the radiation environment around the planets.

The magnetic fields generated by planets or their moons interact with charged particles and can trap them in finite space around the planet. These regions around Earth are known as Van Allen radiation belts. The complex interaction between the solar plasma and the Earth's magnetic field is given as an example in Figure 1. Due to that, interactions with various plasmaenvironments are formed around the Earth. Examples are the electron and proton radiation belts, and the auroras at the pole regions. The boundaries of these regions change dynamically with the change in the intensity of the solar wind and radiation, e.g., due to solar proton events.

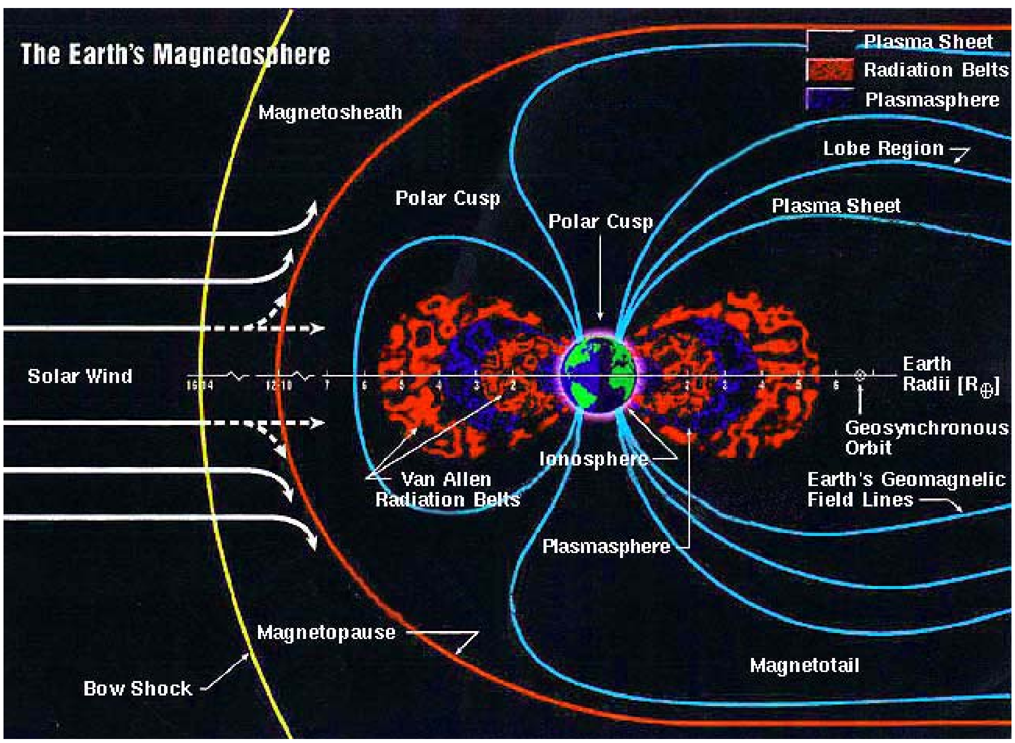

Figure 1. The interaction between the solar wind and radiation and the Earth's magnetic field creates diverse radiation environments, such as trapped particles (e^- and p^+) in the Van Allen belts and impinging radiation in the polar areas.

Figure 1 demonstrates that the plasma environments encountered by earth-orbiting satellites can differ greatly, depending on altitude and inclination. The generalization below is made based on actual calculations of numerous mission conditions, considering the use of 100 mil-thick Al box shielding. Due to the strong dose dependence on the shielding geometry, the reported here values should be taken only as a rough estimate.

1.1. Low Earth Orbit (LEO) and Polar Earth Orbit (PEO)

LEO refers to orbits in the 100-1,000 km altitude range, which includes Earth-observing satellites (EOS). Special case is the Space Station (SS) at ~500 km. The environment in LEO is fairly benign, with a typical dose rate of ~0.1 krad/year. For a mission with a typical duration of 3-5 years, the total dose is <0.5 krad. Most PEO are LEO at high inclination (>55º), but some PEO may be more elliptic. The high inclination takes the orbit through the polar aurora regions, which can be rich ion cosmic ray and solar flare particles. Higher radiation dose is accumulated during the passage through these regions; however, the transition time is typically small in comparison to a full orbit time. Thus, the dose rate is a few krad/year, similar to that at LEO.

1.2. Geosynchronous Earth Orbit (GEO)

GEO is located at 36,000 km altitude in the high-energy plasma sheet. GEO satellites are exposed to the outer radiation belts, solar flares and cosmic rays. The dose rate is of the order of 10 krad/year. For a typical 10 year mission, the total dose is ~100 krad.

1.3. Medium Earth Orbit (MEO)

The radiation environment at MEO (1,000-36,000 km altitude) is harsh since the satellite trajectories are confined mostly within the Van Allen radiation belts. Satellites are usually positioned in regions with somewhat lower though variable particle density between the belts. The dose rate from both protons and electrons can be of the order of 100 krad/year, and it is highly variable due to strong solar cycle effects. Because of that, MEO is only used if no other alternatives exist (e.g., GPS constellation).

1.4. Highly-Elliptic (HEO) and Geo-Tansfer Orbits (GTO)

The conditions at HEO and GTO vary significantly with altitude, eccentricity and the details of the exact trajectory. Spacecrafts in these orbits can cross LEO, PEO, MEO, and GEO, and may encounter additionally higher energy radiation above the poles during solar proton events. A vaguely defined value for the radiation dose rate for HEO is of the order of 10 krad/year. GTO expose a spacecraft to a dose of <5 krad per transit.

1.5. Mars Orbit and Surface

Mars missions, both orbit and surface, acquire a major part of the radiation (~2 krad) before their arrival at Mars. The radiation on the Martian surface is weak (<5 krad/year); thus, little additional dose would be acquired. Due to the weak magnetic field generated by Mars, there are no radiation belts as at the Earth. Orbiters can acquire a total of ~5 krad dose for the duration of the mission. Solar flares are the major concern in transit, in Mars orbit, and on the surface.

1.6. Jupiter Transfer Orbits (JTO) and Europa

Jupiter generates a strong magnetic field, and the radiation environments there is the most intense in the solar system away from the Sun. JTO are used for missions operating in the Jovian environment, as well as for gravitational assist for missions to the outer planets, comets, and deep space. There can be little generalization of the dose rate, since radiation can very by several orders of magnitude along different flight trajectories, ranging from 1 Mrad to 100 Mrad per TO. The dose rate on Europa's surface is of the order of 50 Mrad/year (behind 100 mils Al box shielding).

Missions to Saturn, Uranus, and Neptune must also consider the respective radiation belts, despite these planets weaker magnetic field and larger distance from the Sun (less incident solar particle flux).

1.7. Deep space, comets

Outside the reach of the planet-dominated radiation environments, a deep-space probe is exposed predominantly to galactic cosmic rays. Their intensity is weak, and a dose rate of less than 20 krad/year can be considered typical for deep space. The dose in outer space depends on the flux of the solar wind particles, which falls as $1/r^2$ with the distance from the Sun. Anomalous Galactic cosmic rays also contribute to the acquired dose.

1.8. Man-made: radioisotope thermoelectric generators (RTG)

In addition to the external radiation, some spacecrafts (e.g., deep-space missions, long-duration Mars missions) may require the use of a RTG. Although the positioning of the RTG, and the used shielding, will strongly influence the dose to the different devices, in general, the addition to the total dose is small (<2 krad) and often negligible. However, RTGs produce neutrons, which cause displacement damage in materials. Their flux falls with the distance from the source with the inverse-square law.

2. Typical performance by technology

The performance of electronic components varies by device technology, feature size (e.g., transistor gate length, also referred to as technology node), architecture, and design. *Table I* presents the typical performance of device categories in various radiation environments. It should be stressed, that there are known examples of devices with identical function and parameters built by different manufacturers, whose performance in radiation environments can differ greatly. *Table I* reflects the characteristics of the technologies, rather than manufacturer-related characteristics.

Table I does not reflect the dependence of the performances as a function of the advancing technology. For example, as a consequence of the decreasing feature size, the reduced volume of the gate dielectric of a Si transistor is less affected by the total ionization dose. The exact opposite trend is in place for the single-event effects, due to the higher charge density (in the smaller active volume) created by the radiation. *Table I* can be considered representative for the 0.18 μm and 2.0 μm technology generations.

The information in *Table I* refers to electronic components. For assemblies, subsystems and systems, analysis should be done with consideration of the application. Potentially severe requirements are imposed if the performance of the most sensitive part is attributed to the whole assembly.

Table I *Typical performance of electronic components in radiation environments*

Legend:

green	Expected normal function
yellow	Assessment needed
orange	Undefined performance
red	Special measures required

		Radiation Environments													
		TID [krad(Si)]							Displacement Damage (1 MeV n^0 / cm^{-2} equivalent)				SEE		ELDRS
		~0.1	<1	2-5	10	20	100	>1000	5×10^9	2×10^{10}	2×10^{11}	10^9-10^{11}	All Environments		
		1yr. LEO	1yr. PEO	1yr. Mars (orbit/surface)	1yr. GEO / HEO, 1 GTO	1yr. DS	1yr. MEO, 1 Jupiter TO	1yr. Jovian (Europa)	LEO, DS, Mars	PEO, GEO	MEO, Jupiter	Man-made RTG (additive)	Recoverable Events SEU, SET	Permanent Events SEL, SEGR, SEB	
CMOS	Linear	green	orange	orange	red	red	red	red	green	green	green	green	orange	orange	green
	Mixed Signal	green	yellow	yellow	red	red	red	red	green	green	green	green	orange	orange	green
	Flash Memory, DRAM	green	yellow	yellow	red	red	red	red	green	green	green	green	orange	orange	green
	SRAM	green	yellow	yellow	orange	orange	red	red	green	green	green	green	orange	orange	green
	Digital Logic	green	yellow	yellow	yellow	yellow	red	red	green	green	green	green	orange	orange	green
	Microprocessor	green	yellow	yellow	orange	orange	red	red	green	green	green	green	orange	orange	green
BiCMOS Linear		green	orange	orange	red	red	red	red	green	green	green	green	orange	green	orange
Bipolar	Mixed Signal	green	green	green	green	green	yellow	yellow	green	green	green	green	orange	green	green
	Standard Linear	green	yellow	yellow	yellow	red	red	red	green	green	green	green	orange	green	orange
	Digital	green	green	green	green	green	yellow	yellow	green	green	green	green	orange	green	green
Power MOSFET		green	green	yellow	yellow	orange	orange	orange	green	green	green	green	orange	orange	green
JFET		green	green	green	green	green	yellow	yellow	green	green	green	green	green	green	green
BJT	Power	green	green	green	green	orange	orange	orange	yellow	yellow	red	orange	green	green	green
	Signal	green	green	green	green	green	orange	orange	green	yellow	orange	orange	green	green	green
SOI		green	green	green	green	green	yellow	yellow	green	green	green	green	orange	green	green
SiGe RF		green	green	green	green	green	yellow	yellow	green	green	green	green	orange	green	green
III-V Electrical	SRAM	green	green	green	green	green	green	green	green	green	yellow	yellow	orange	green	green
	RF (transistors, diodes)	green	green	green	green	green	green	green	green	green	yellow	yellow	orange	green	green
III-V El.-Optical	Lasers, LED	green	green	green	green	green	green	yellow	yellow	orange	orange	orange	orange	green	green
	Detectors, solar cells	green	green	green	green	yellow	yellow	yellow	yellow	orange	orange	orange	orange	green	green

3. Performance characterization tests

Parts performance in radiation environments is assessed from testing Total Ionizing Dose (TID), Displacement Damage (DD) and Single Event Effect (SEE) sensitivity, relative to the expected ranges shown in *Table I*. The tests should be done in accordance with MIL-STD-883 or equivalent. In addition to these tests, radiation lot acceptance test (RLAT) needs to be performed for parts, which are not fabricated on a radiation-hardened process, or are not inherently immune to radiation. This relates to the vast majority of all commercially available parts. The criterion for this is the successful demonstration that the device is capable of surviving 3x or greater the radiation levels in *Table I*. Typically, radiation data should demonstrate 90% confidence that the population probability of survivability is at least 99%.

3.1. Total ionizing dose (TID) level

Ionizing radiation looses energy primarily by creating electron-hole pairs, and is associated with breaking bonds and creating point defects, which accumulate with radiation exposure. Defects in the active device regions trap charge, and thus affect its performance characteristics. Above a given defect concentration, a device seizes to function. This defines the TID level for that device.

The successful use of electronic devices in radiation environments requires that, all flight parts must operate within post-irradiation specification limits following exposure to twice (2x) the expected total dose environment specified in *Table I*. The factor of 2 accounts for the uncertainty in the radiation environment. The lot-to-lot variation and the uncertainty from the radiation experiments must be factored additionally, if significant. The TID radiation environment includes all radiation components, X-rays, gamma rays, protons, electrons, and heavy ions.

3.2. Displacement damage (DD)

Displacement damage characterizes the vacancy-like defects created as atoms are knocked out of their positions by incident ions or recoils. TID-tolerant devices can be sensitive to DD. Potentially susceptible parts include, but are not limited to, optical devices, photo-detectors, charge-coupled devices, optocouplers, LEDs, laser diodes and precision bipolar linear devices. Therefore, DD susceptibility assessment is needed as a complementary tool. All devices must operate within specification limits following exposure to twice (2x) the expected environment specified in *Table I.*

3.3. Recoverable single events: single event upset (SEU)

The energy of an ionizing particle deposited per unit length (l) in a given material is used to define the linear energy transfer (LET). LET is measured in units of $MeV \times cm^2/mg$, and reflects the amount of charge generated by the incident particle for the considered material. LET is used to characterize the performance of microcircuits containing bi-stable elements (e.g. flip-flops, counters, RAMs, microprocessors, etc.) in radiation environments in terms of SEU. The charge (electron and holes), generated by the ionizing radiation in the active region of a device, is driven by the electric fields to the electrodes. This creates currents, which can be sufficiently large to change a bi-stable element, e.g., switch a transistor, the effects of which can propagate to change the status of a system. A false SEU-generated signal is indistinguishable from an actual event.

The importance of an SEU event to a system performance or a mission success varies significantly. For example, a loss of one bit in an image can be ignored, whereas a bit-flip sending a wrong command (e.g., to fire engine) can lead to spacecraft loss.

Thus all microelectronic devices containing bi-stable elements should be characterized for SEU with respect to their function, so that an upset rate calculation can be performed. Typically, the characterization of electronic devices is extended to a fluence of 10^7 ions/cm², and typical requirements for SEU acceptability are:

- No upsets observed during SEU testing to an LET of 75 MeV×cm^2/mg, or
- Verification of device bit error rate of 10^{-10} per day or better in the galactic cosmic ray environment, or
- Calculation of a device's upset rate shall be equal to or less than the required circuit upset rate as determined by circuit SEU analysis.

3.4. Non-recoverable (permanent) single events

Depending on device design and/or architecture, in some cases the incidence of a single ion in the active device region can create irreversible effects, which can render a system inoperable.

3.4.1. Single Event Latchup (SEL)

The ionization tracks formed by ions can create channels for charge current to flow through parasitic transistors, inherent for most CMOS circuits. The device architecture is such that these currents are amplified as in a silicon rectifier, and cannot be terminated until power is removed. This process is referred to as SEL. These large localized currents can lead to instant (e.g., wire melt) or latent (e.g., electromigration) failures. Latchup occurs in 20-50 ns, significantly slower than SEU, and the result is sometimes non-catastrophic.

All CMOS devices (including those with epitaxial layers) should be subject to latchup evaluation and should be tested to a fluence of 10^7 ions/cm². Exception can be made for silicon-on-insulator (SOI) CMOS, whose design eliminates the parasitic bipolar transistors. The beam angle must not exceed 60 degrees and test ions must have an effective range greater than 35 microns. Bias must be at specified maximum voltage. Tests must be performed at room ambient and at elevated temperature of 125°C or the maximum specified operating temperature of the part. SEL is difficult to characterize; however, typical requirements for SEL susceptibility are:

- No latchup to an LET of 75 MeV×cm^2/mg, or
- Verification that the device latchup probability in the mission environment (*Table I*) be less than 10^{-4} /device-year for parts that exhibit latchup between 35 MeV×cm^2/mg and 75 MeV×cm^2/mg.

3.4.2. Single Event Gate Rupture (SEGR)

SEGR is often trigerred by heavy ions, when the generated charge is transported to regions near gate dielectrics. The induced electric fields across the dielectric can exceed significantly its dielectric breakdown field, which always results in a catastrophic failure. Most sensitive to SEGR are power devices, programmable devices, and devices with very thin oxide layer. In general, the devices sensitive to SEGR are also sensitive to electrostatic discharge and electrical overstress.

Devices, especially power MOSFETs operated in the off-mode, should be evaluated for SEGR at the worst-case application V_{GS}. Testing is usually performed with normal beam incidence and at room ambient temperature. The survival voltage (V_{DS}) is established from exposure to a minimum fluence of 10^6 ions/cm² of an ion with a LET $\geq$ 37 MeV×cm^2/mg and with a range greater than 150% of the depletion depth. To avoid SEGR, the application voltage should be derated to 75% of the established survival voltage.

3.4.3. Single Event Burnout (SEB)

SEB is a problem, which is often associated with power transistors (MOSFET or bipolar). In this case, a large localized current flows in the body of the device that exceeds the normal current density within the structure, causing localized melting of the Si along defects or regions with non-uniform doping. SEB results in a permanent failure.

Such power devices should be evaluated for single event burnout (SEB) at the worst-case application V_{BE} (for bipolar devices) or V_{GS} (for MOS devices). Testing is usually performed with normal beam incidence and at room ambient temperature. The survival voltage (V_{CE} or V_{DS}) is established from exposure to a minimum fluence of 10^6 ions/cm² of an ion with a minimum LET of 37 MeV×cm^2/mg and with a range greater than 150% of the depletion depth. Test requirements for SEB are similar to those for SEGR except that the drain current (MOSFET) or collector current (bipolar transistor) must be measured to determine if burnout occurs. To avoid SEB, the application voltage should be derated to 75% of the established survival voltage.

3.5. Enhanced Low Dose Rate Sensitivity (ELDRS)

Some types of bipolar transistors are damaged to a much larger degree when irradiated at a low dose rates, compared to damage that occurs when irradiated at high dose rates (>100 rad(Si)/s), which are usually used for characterization. This effect is known as ELDRS. The dose rate in most space radiation environments is significantly lower than the dose rate in any ground test; thus, if ELDRS is ignored, flight operation of such bipolar transistors may deviate significantly from the expected performance.

Thus, all linear bipolar and BiCMOS ICs should be evaluated for susceptibility to ELDRS. The sensitivity range for ELDRS requires that tests should be performed for dose rates greater than 0.005 rad(Si)/s and less than 10 rad(Si)/s. Parts are typically exposed to three times (3x) the expected TID environment (*Table I*). Parametric degradation due to ELDRS should be accounted for in the circuit worst-case analysis.

III. THERMAL ENVIRONMENT

1. Definition of thermal environments

Electronic devices and assemblies are usually enclosed in a controlled thermal environment in the spacecraft interior, provided by a high-thermal capacity base plate. The base plate temperature is determined by the external heat absorbed by the spacecraft and the heat generated by the functioning electronic components, and is regulated by passive heat distribution and active heating elements. Spacecraft may receive radiant thermal energy from three natural sources:

- incoming solar radiation (solar constant): 0–1353 W/m^2;
- reflected solar energy (albedo): 0–0.32 of the solar radiation; 0–450 W/m^2 global annual mean;
- outgoing long-wave IR radiation emitted by the Earth and atmosphere: 100–270 W/m^2.

These values refer to an Earth orbiter. Averaged over long time, the Earth and its atmosphere is in a radiative equilibrium with the Sun. However, it is not in balance everywhere on the globe and there are important variations with local time, geography, and atmospheric conditions. The local and/or momentary variations are seen in LEO, PEO and MEO orbits. The reflected and outgoing radiation terms are negligible above 4 Earth radii, thereby for GEO orbits.

1.1. <u>Spacecraft interior (controlled)</u>

The thermal environment relevant to electronic components interior to the spacecraft is controlled within narrow margins, typically 5-10 ºC for an Earth orbiter, and 15-20 ºC for a Mars orbiter. Figure 2 shows a typical example of the measured temperature at the base plate of a satellite in LEO. However, the thermal requirements are extended to a broader range to guarantee that the components will function after accidental loss of power or overheating. These define the worst-case high and low temperature limits, shown in Figure 3 for several missions. The –10 ºC and +55 ºC limits envelope all of these missions, and can be regarded as a typical environment for the interior of orbiting satellites and spacecraft in general.

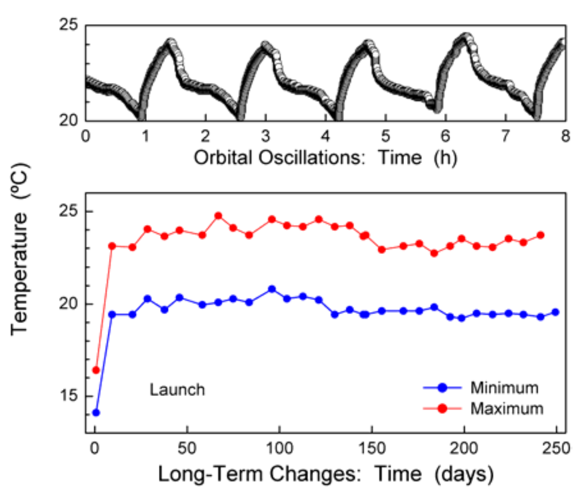

Figure 2. Short- (top) and long-term (bottom) temperature oscillations at the base plate of LEO orbiting satellite.

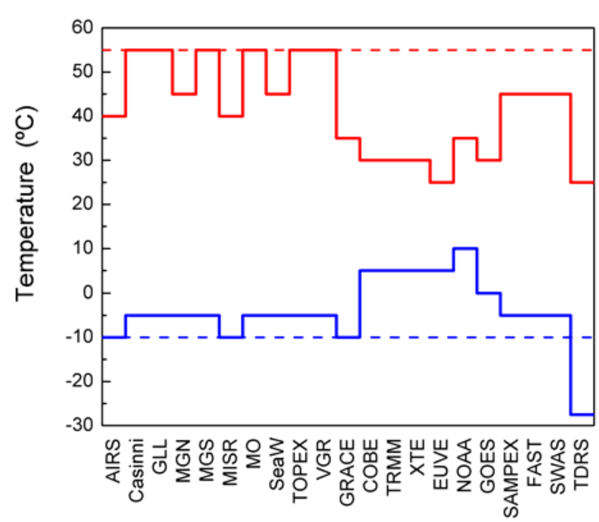

Figure 3. Worst-case high (red) and low (blue) temperatures for several missions. The dashed lines at –10 ºC and +55 ºC envelope all missions with the exception of TDRS operating at lower T.

1.2. Spacecraft exterior (exposed to space)

The spacecraft exterior can be exposed to direct thermal radiation as well as to the cold space. This results in extreme temperatures anywhere from –120 ºC to +150 ºC. These environments are relevant to solar arrays, and to external sensors and components of electronic apparatus. Orbiting spacecraft can thus be exposed to extreme thermal cycles. The thermal conditions for small objects, positioned close to the satellite body, tend to be determined by the satellite temperature (through its size, mass, and thermal inertia). For objects with large area and small mass (e.g., solar sails, solar panels, etc.), the solar radiation, albedo, and outgoing IR radiation around the planet dominates the thermal environment.

1.3. Landers and probes

Landers and probes can encounter the largest variation of thermal environments. Some examples are: extremely hot (Venus, Sun), extremely cold (Titan, Europa), and extreme thermal cycles (Mercury). Engineering a suitable thermal control is possible in most cases; however, it comes at high cost. Passive thermal shielding adds extra mass, whereas thermal heating affects the power budget. Both mass and power are limiting resources in all space exploration missions. Therefore, every environment of interest must be considered on a case-by-case basis.

1.3.1. Mars lander

A Mars lander can have a partially controlled environment, protecting some of the electronic components, while others are exposed to the surface Martian conditions.

a) *Interior (partially controlled)*

The worst-case temperatures range for on-board electronics is typically from –40 ºC to +50 ºC. The temperature can vary between these limits with the cycle of the Martian day (sol), which imposes more stringent thermal conditions on the electronic devices. Current missions have a duration of ~90 sols, but long-term missions (~500 sols) are being planned. The requirements for future missions with longer duration will approach these for exterior electronics.

b) *Exterior (exposed to planetary surface environment)*

Exterior electronic devices and assemblies are exposed to the Martian surface conditions. The minimum and maximum daily temperatures are –125 ºC and +25 ºC, respectively. Devices, which retains their functional characteristics at these conditions, minimize thermal management requirements.

1.3.2. Venus lander, solar probe

Venus hosts a hot and highly chemically reactive environment with a nearly constant temperature averaging at approximately +470 ºC at the planet surface. At the present, there is no electronic device technology capable of producing functional parts at these conditions. Thermal protection must be engineered through the implementation of cooling mechanisms, and even then the electronic components cannot survive for a long time. Past Venus surface missions have survived typically ~1 hour (except one two-hour-long mission). Some atmospheric layers, which can be targeted by sample return missions, can be significantly cooler; as low as ~70 ºC.

The thermal environment for a solar probe is determined by the solar radiation, based on the proximity to the Sun. High-temperature environments can approximate these on Venus.

1.3.3. Europa lander

A surface mission on Europa will encounter a –145 ºC average temperature, but no knowledge exists for the moon's interior. It is believed that under the thick surface ice lies liquid water, where the thermal conditions may be more Earth-like. The robot under consideration to explore sea depths may encounter much warmer conditions near heat sources such as thermal vents. Alternatively, the planet interior may contain softer flowable ice (analogous to glaciers).

1.3.4. Mercury Orbit and Surface

The Mercury surface, with its daily temperatures varying between –180 ºC and +425 ºC, provides extreme testing conditions for electronics. Orbiting spacecraft will also experience large amplitude thermal cycles, while some constantly shadowed regions on the surface are attractive for a landing mission.

2. Typical device performance

The typical device performance per technology is shown in *Table II*. It should be stressed that the table is less representative of the technology itself, but it reflects the degree of complexity of the typical devices. For example, extremely low temperatures do not impede CMOS technology itself, but complex CMOS devices can lose functionality due to loss of drive current and timing correlation among operations.

3. Performance characterization tests

The characterization thermal tests can vary significantly depending on the use of a devices, and not only on the device type and technology. For example, a deviation of a functional parameter resulting from extreme temperature may be critical in one application, and unimportant in another. The recommendations below are designed to assess principal functionality, and does not address application-specific details. It is recommended that characterization tests are performed in accordance with MIL-STD-883, or equivalent, for temperatures encompassing the ranges in *Table II*.

3.1. Thermal vacuum

The heat generated by the operation of electronic devices is dissipated, in order of importance, by conduction, convection, and radiation. Maintaining good heat conduction is essential for electronic components in space, since convection is eliminated in the transition from atmospheric pressure to vacuum, and heat radiation is highly ineffective. The operation of some devices, such as MEMS or box-packaged electronics, can be affected by the loss of pressure. In such cases, thermal vacuum tests must be performed. Conventionally packaged electronic devices need not be subjected to thermal vacuum tests.

3.2. Extreme temperatures

The performance of all electronic devices must be characterized throughout the temperature ranges given in *Table II*. It must be demonstrated that all electronic components operate within the specified parameters at the extreme temperatures for each environment, including continuous operation and starts. In case parametric deviations are observed, judgment can be made on a case-by-case basis, considering the critical performance parameters for the specific application.

3.3. Thermal cycling

The thermal cycling requirements for flight can also vary significantly with respect to part operations. At least three (3) cycles are recommended at the board level in order to uncover major defects. To test parametric functionality, a general rule is to perform three times (3x) the actual number of thermal cycles planned for a mission. These cycles are performed on a qualification model.

Table II. *Typical performance of electronic device technologies in thermal environments*
Legend: *green – major effects unlikely;*
yellow – failures possible / assessment needed;
orange – undefined performance / special measures may be required;
red – special measures required.

		Thermal Environments					
		Orbiter / Deep Space		Lander / Probe			
				Mars		Venus	Europa
		Interior (–10ºC/+50ºC)	Exterior (–120ºC/+150ºC)	Interior (–40ºC/+50ºC)	Exterior (–125ºC/+25ºC)	+475ºC	–145ºC
CMOS	Linear	green	orange	orange	orange	red	orange
	Analog RF	green	orange	orange	orange	red	orange
	Mixed Signal	green	orange	orange	orange	red	orange
	Digital Logic	green	yellow	yellow	orange	red	orange
	Flash Memory	green	yellow	yellow	orange	red	orange
	Processors	green	yellow	yellow	orange	red	orange
BiCMOS	Linear	green	orange	orange	orange	red	orange
Bipolar	Compliment.	green	yellow	yellow	orange	red	orange
	Linear	green	orange	orange	orange	red	orange
	Digital	green	yellow	yellow	orange	red	orange
MOSFET		green	yellow	yellow	orange	red	orange
JFET		green	orange	orange	orange	red	orange
BJT	Power	green	yellow	yellow	orange	red	orange
	Signal	green	yellow	yellow	orange	red	orange
SOI		green	yellow	yellow	orange	red	orange
SiGe		green	yellow	yellow	orange	red	orange
III-V Electr.	SRAM	green	green	green	green	red	green
	RF	green	green	green	green	red	green
III-V El.-Opt.	Laser, LED	green	yellow	yellow	yellow	red	yellow
	Detect., solar cell	green	yellow	yellow	yellow	red	yellow

IV. VIBRATION ENVIRONMENT

1. Definition of vibration environment

Flight hardware is exposed to vibrations during launch and during mission duration. The launch vibrations originate from engine ignition and operations, atmospheric drag, and stage separations. Trajectory corrections using on-board engines cause vibrations while in orbit.

Three different categories are used to describe the vibration environment – acoustic vibrations, random vibrations, and pyroshock. Acoustic noise is the major source of vibrations. Each category is determined by the launch vehicle used, the payload mass and configuration, and by their mechanical coupling. Specialized software uses this information to estimate the payload performance in given vibration environments.

1.1. Acoustic vibrations

Acoustic energy is the primary source of vibration input to a space launch vehicle. During the initial phases of a rocket launch, high velocity gases are ejected from motor nozzles and reflected from the ground, creating turbulence in the surrounding air and inducing a vibratory response of the rocket structure. During the subsequent ascent phase of a launch, as the vehicle accelerates through the atmosphere to high velocity, aerodynamic turbulence induces pressure fluctuations which again cause structural vibration. These pressure fluctuations increase in severity as the vehicle approaches and passes through the speed of sound, due to the development and instability of local shock waves. The high-level acoustic noise environment continues during supersonic flight, generally until the maximum dynamic pressure, or max Q, condition is reached.

Acoustic energy gets transmitted to the mission payload in two ways. First, fluctuating pressures within the payload fairing impinge directly on exposed spacecraft surfaces, inducing vibration in high gain antennae, solar panels and other components having a large ratio of area-to-mass. Secondarily, the fluctuating external pressure field causes an oscillatory response of the rocket structure, which is ultimately transmitted through the spacecraft attachment ring in the form of random vibration. From the spacecraft perspective, this random input is generally lowest at the launch vehicle attachment plane, and increases upward along the payload axis.

Figure 4 shows a 2σ (95%) envelope, represented by the solid line and the tolerance (dashed lines) of the acoustic noise level from 18 recent JPL missions. The pressure P is measured in decibels, defined as $dB = 20 \cdot \log\left(P/P_{ref}\right)$, where $P_{ref} = 2\times10^{-5}$ Pa is the audible limit of the human ear.

1.2. Random vibrations

The random vibration environment consists of stochastic instantaneous accelerations, which are input to a microelectronic component or assembly, transmitted via spacecraft structure under launch dynamic excitation conditions. Random vibration input occurs over a broad frequency range, from about 10 Hz to 2000 Hz. In the space vehicle launch environment, random vibration is caused primarily by acoustic noise in the payload fairing, which is in turn induced by external aerodynamic forces due to dynamic pressure and reflection of rocket exhaust from the ground. Due to the large diversity of linear dimensions and component masses exposed to the initial vibrations, as well as the different Q-factors, the random vibration spectrum appears to be continuous. The dominant part of it is contained in the 20-2000 Hz range (Figure 5).

It should be noted that the random vibration spectrum depends on both, launch vehicle and payload. The launch vehicle spectra (Figure 5, open symbols) give the initial vibration conditions. The payload contribution depends on the equipment mass. Figure 5 shows the 95% envelope (margins: dashed lines) of the random vibration spectrum on a 5 kg unit.

1.3. Pyrotechnic shock

Pyrotechnic shock is a design and test condition under which flight hardware is subjected to a rapid transfer of energy. The energy transfer is associated with the firing of an explosive device, usually for the purpose of initiating or performing a mechanical action. Spacecraft separation events or the release of propulsion system safing devices are typical such mechanical actions. Pyrotechnic shock also occurs in flight during engine firing for orbit correction.

The release of energy from an ordnance-containing device and the subsequent transfer to the surrounding structure represents a very complex event. As a result, it is difficult to describe the actual shape of the applied shock wave; it is generally not a simple time-based pulse such as a square or triangular wave. Figure 6 shows the 2σ pyroshock environment for several typical missions encompassing 95% the full range of analyzed data. For test purposes, this environment could be considered a qualification level.

2. Typical device performance

Monolithic electronic devices have small physical dimensions and mass, which makes them generally insensitive to vibrations. All devices, listed in *Table I*, are expected to perform within their specifications. Because of that, vibration tests are rarely performed on monolithic components.

Some assemblies, such as hybrids, are occasionally susceptible to electrical connection or physical failures under vibration. It is recommended that they are tested prior to integration into a board assembly. Board assemblies, subsystems and systems are susceptible to failure, and tests are necessary. In some cases, redesign or other special measures may be required to qualify the equipment according to mission-specific requirements.

3. Performance characterization tests

Device performance with respect to vibration environments can be done in accordance with MIL-STD-1540B, or equivalent.

3.1. Sinusoidal test

Sinusoidal vibration is employed to simulate the effects of significant flight environment launch transients. These transients typically produce the dominant loading on primary and secondary structure and many of the larger subsystems and assemblies. Sinusoidal vibration is the only widespread current method of adequately exciting the lower frequency dynamic modes, particularly those below 40 Hz. However, it should be noted that a high-load sinusoidal test can significantly overstress the structure. Excessive fatigue cycles can be avoided by sweeping at a log rate between 1 octave/minute and 6 octaves/minute. The higher rate is near the upper limit, which most control systems can accommodate without experiencing some instability. The use of logarithmic sweep rates has the advantage in that a nearly equal time is spent at resonance for a given Q, independent of frequency.

3.2. Acoustic vibration test

The fluctuating pressures associated with acoustic energy can cause vibration of structural components over a broad frequency band, ranging from about 20 Hz to 10,000 Hz and above. Such high frequency vibration can lead to rapid structural fatigue. Thus, the objective of a spacecraft acoustic noise requirement is to ensure structural integrity of the vehicle and its components in the vibroacoustic environment.

An 95%-envelope acoustic specification of 18 recent JPL missions for a variety of payloads and different launch vehicles is shown in Figure 4. The solid line represents the requirement; the tolerances are given by the broken lines. The typical recommendation is that equipment should be able to withstand at least 3 minutes during test in the acoustic requirement of Figure 4. This requirement considers the STS, Proton, Taurus, Ariane and Titan launch vehicles, except the most recent and more powerful launch vehicles.

3.3. Random vibration test

Random vibration criteria should be developed by the process described as follows:

A. Determine the power spectral density (PSD) of the random vibration directly transmitted into the flight article through its mounts from the launch vehicle sources such as engine firing, turbopumps, etc. These vibration conditions at the launch vehicle-to-payload interface are typically available from the launch vehicle developer.
B. Perform an analysis to predict the payload/flight article's vibration response to the launch vibro-acoustic environment, using statistical energy analysis (SEA) methods. For example, VAPEPS (Vibro-Acoustic Payload Environment Prediction System) is effective in the higher frequencies, and NASTRAN is effective in the lower frequencies range.
C. Establish a minimum level of vibration, which is necessary to ferret out existing workmanship defects and potential failures.
D. Envelope the curves from steps 1-3 to produce a composite random vibration specification for the test article.

The 95% envelope requirement shown in Figure 5 is described by a +6 db/octave slope from 20-50 Hz, a weight-dependent constant PSD level, and a –4.5 db/octave slope from 500-2000 Hz. The PSD level depends on the weight (w in kg) as follows:

$$PSD(w) = 0.1 \times \frac{w+20}{w+1} \qquad g^2 / Hz$$

Launch random vibration tests are generally applied in each of three orthogonal axes, and have a Gaussian distribution of the instantaneous acceleration. Both the Acceleration Spectral Density and wideband acceleration should be within specified tolerances. Each assembly or subsystem should be in its launch configuration, attached to vibration test fixtures at their normal flight structural interfaces. The duration of the random vibration test is typically 3 minutes.

3.4. Pyrotechnic shock

The requirement in Figure 6 (solid line) and the tolerance (± 6 dB; broken lines) is a 95% envelope of 9 missions. The slope up to 1500 Hz is 9 dB/octave. Above 1500 Hz, the pyroshock response is equal to 4000 g. A Q-factor of 10 is considered for this analysis. Equipment should be exposed 3 times in each axis and in each direction to the shock requirement in Figure 6. For devices with self-contained ordinance, 3 self-induced shocks should also be applied.

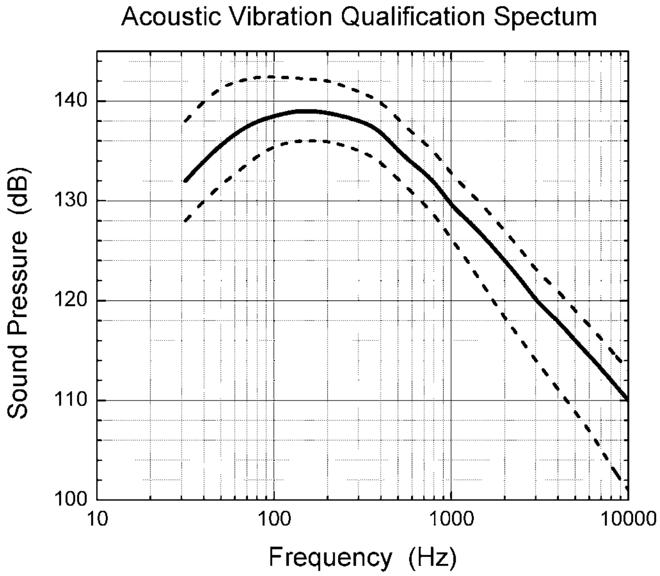

Figure 4. A 95% envelope of the acoustic environment of 18 recent JPL missions. Solid line is the requirement, dashed lines – tolerance.

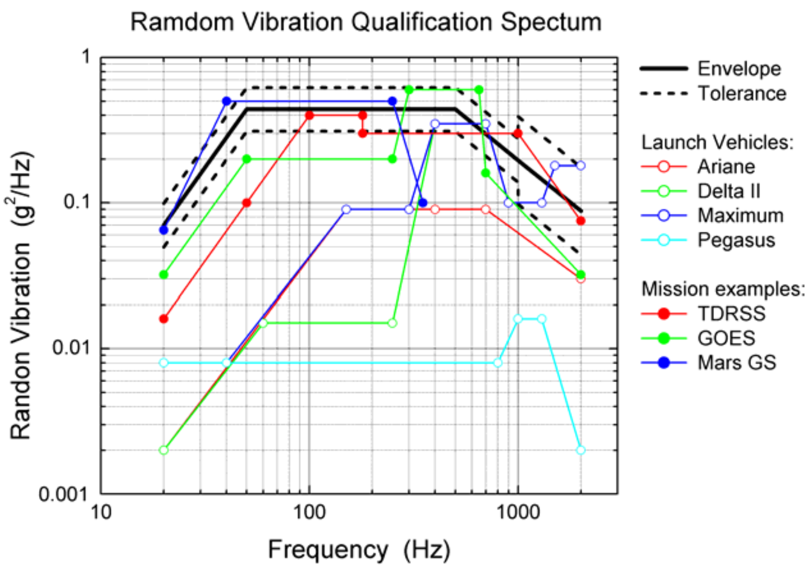

Figure 5. A 95% envelope of the random vibration environments of 18 JPL missions. Open symbols refer to launch vehicle profiles. Solid symbols give actual mission data. The solid line is the 95% envelope for a 5 kg unit.
Note: The actual random vibration spectrum depends on the unit mass and the coupling between payload and launch vehicle.

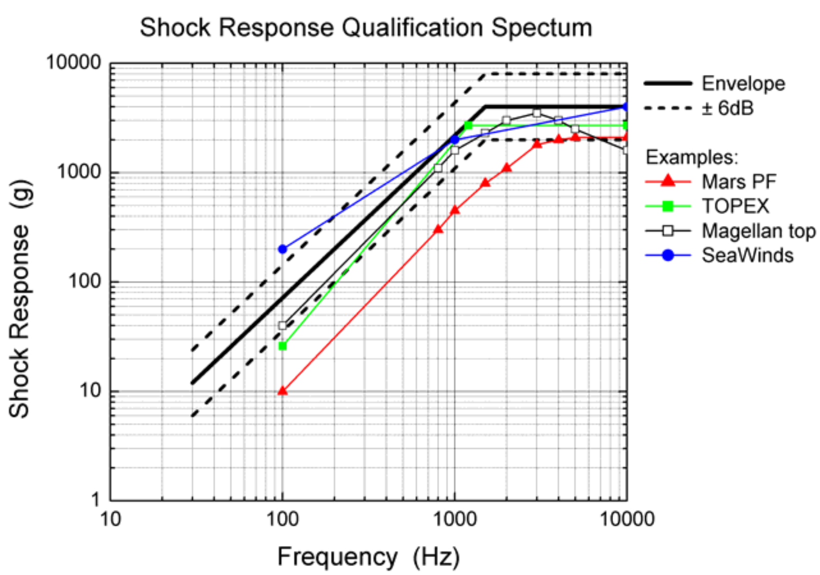

Figure 6. A 95% envelope of the pyrotechnic shock environments of 9 JPL missions. Solid line marks the requirement, dashed line – the tolerance. The four missions shown for reference (symbols) encompass the minimum and maximum environments of all considered missions.

V. ELECTROMAGNETIC / ELECTROSTATIC / ELECTRICAL ENVIRONMENT

1. Definition of electromagnetic, electrostatic and electrical environment

1.1. <u>Electrical and electromagnetic impulse (EMI) environment</u>

The electrical environment in satellites and spacecraft varies greatly with the on-board power sources. Other than the commonly used solar arrays, power sources such as RTGs, solar-electric propulsion, etc., may impose different requirements on the bus power. Even for solar array power there is no established standard. Solar arrays produce typically between 100 V and 160 V, which is sometimes used as bus potential in commercial satellites. For most NASA missions, the bus voltage is 28 V DC (min: 22V; max 36V). The bus voltage is regulated to different board requirements (e.g., ±15V, ±10V, ±5V, ±3.3V). Regulators are typically used on each board to provide power for individual components (e.g., ±5V, ±3.3V). The load requirement is considered in the regulation of the dynamics (spikes, power interrupts, etc.) in accordance with MIL-STD-461 and MIL-STD-462 or equivalent. This standard regulates the electromagnetic emissions and susceptibility for the control of electromagnetic interference, discussed below.

The typical EMI requirements are shown in Figure 7. (The feature at 2 GHz marks the S-band communication frequency, around which stricter requirements are imposed.)

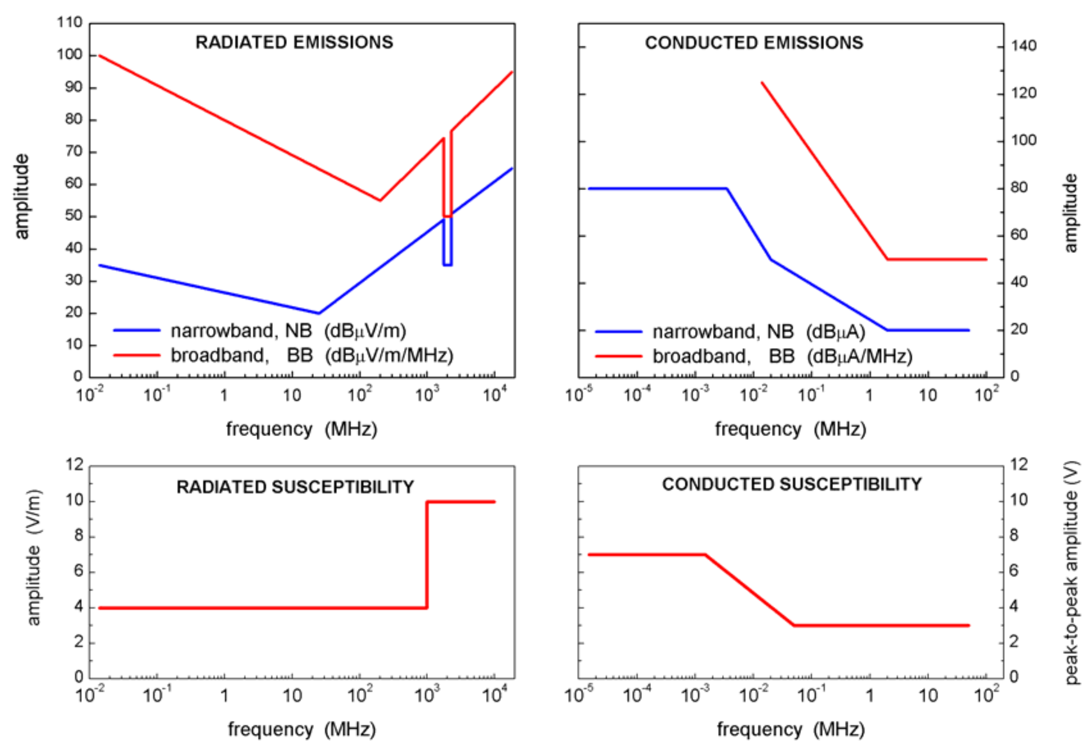

Figure 7. Typical requirements for radiated emissions (top left), conducted emissions (top right), radiated susceptibility (middle left), and conducted susceptibility (middle right).

1.2. Electrostatic environment

The electrostatic environment for parts and assemblies depends on the charging of a spacecraft, to which both the surrounding plasma environment and the spacecraft design contribute. The measures used to mitigate the risks of permanent damage to electronic components caused by electrostatic discharge (ESD) or electrical overstress (EOS) are not discussed here, as they are design-dependent.

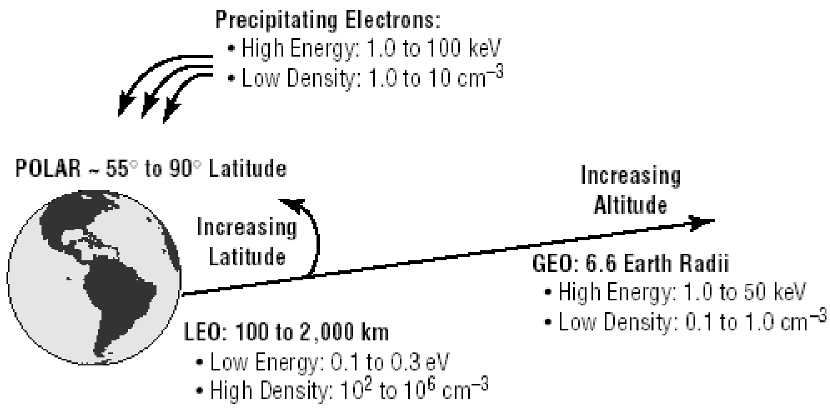

Figure 8. A schematic representation of the plasma environment at LEO, PEO, and GEO.

Common is the plasma environment (Figure 8), which causes spacecraft charging. Near the Earth, the plasma is dense and cold. Farther away, the density drops fast, however, the plasma energy increases out to GEO. The plasma environment is a dynamic one, determined by the interaction of the Earth magnetic field and the solar wind (see Figure 1). Solar flares affect the plasma environment by heating and expanding the boundary of the neutral atmosphere, and by providing energetic particles, which increase the plasma density and temperature. The photo-effect under direct sunlight counteracts the charging by providing an outflow of low-energy electrons. This, however, may create potential differences between shaded and illuminated areas of a dielectric, which can cause ESD. Figure 9 shows the calculated maximum charging potential of a spacecraft in Earth orbit as a function of altitude and latitude.

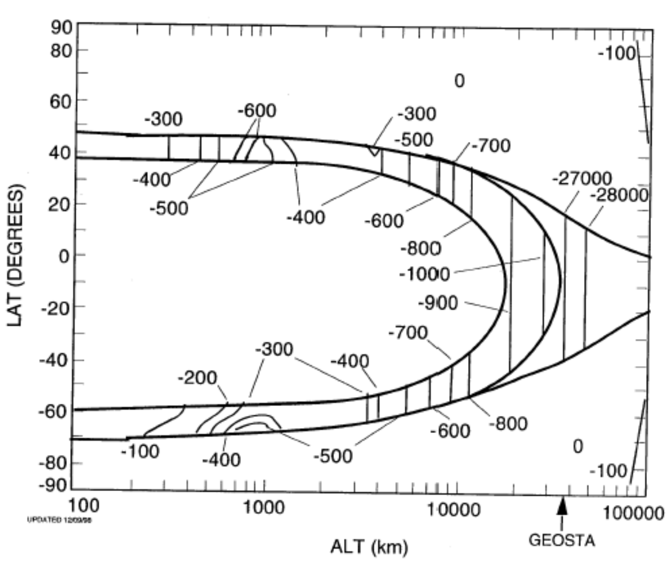

Figure 9. The maximum charging potential of a spacecraft as a function of altitude and latitude.

Satellites in LEO are exposed to cold dense ionospheric plasma. The high plasma density results in a short Debye length, which measures the screening distance of the spacecraft potential, and determines the appropriate charging model. A satellite can charge up to a potential of the order of 10 V. Charging at LEO may have a significant effect on solar arrays, which can charge up to 100-150V.

In addition to the LEO conditions, a satellite at a PEO orbit is exposed to the Auroras over the poles and it is sensitive to solar flares, the protons from which are channeled at the polar regions. Thus, a spacecraft can charge to a potential greater than 100 V.

GEO orbits are located in regions with a high-energy low-density plasma, and are strongly affected by solar proton events. The low plasma density results in a long Debye length. Satellites can charge up to 10,000-25,000 V relative to space.

2. Typical ESD and EMI device performance

Table III. *Typical ESD and EMI performance of electronic device technologies. Legend:*
green – major effects unlikely;
yellow – assessment needed;
orange – undefined performance (special measures may be required).

		EMI / ESD Performance	
		EMI	ESD
CMOS	Linear	green	orange
	Analog RF	green	orange
	Mixed Signal	green	orange
	Digital Logic	green	yellow
	Flash Memory	green	orange
	Processors	green	yellow
BiCMOS Linear		green	orange
Bipolar	Compliment.	green	yellow
	Linear	green	orange
	Digital	green	yellow
MOSFET		green	orange
JFET		green	yellow
BJT	Power	green	green
	Signal	green	yellow
SOI		green	orange
SiGe		green	yellow
III-V Electr.	SRAM	green	yellow
	RF	green	yellow
III-V El.-Opt.	Laser, LED	green	yellow
	Detect., solar cell	green	yellow

The ESD performance cannot be categorized by technology. Consider, for example CMOS technology. With the decrease of the feature size, the ESD sensitivity increases significantly, while the technology itself remains the same. The ESD sensitivity is determined by details of the architecture of the electronic components and the used dielectric materials. In general, it depends on dielectric breakdown field, dielectric thickness, field gradients, and flowing current. When assessing the ESD sensitivity of a device, the key factors to look for are:

- Dielectric thickness – More advanced devices have smaller feature size, thereby thinner dielectric layers. These devices are built for smaller tolerance in voltage variations. They are more sensitive to surges caused, for example, by ESD, which may occur anywhere in the spacecraft and propagate to the device circuit.
- Feature size – devices with smaller feature size are more sensitive to ESD for the reasons discussed above.
- Power devices – Such components are generally less susceptible to ESD damage. They are built to withstand large currents, and have higher tolerance to peak currents caused by ESD-voltage surges.
- Device architecture – For the same feature size, device design is responsible, for example, for the higher ESD sensitivity of CMOS compared to bipolar devices; although this is not always the case if different technology nodes are compared.

3. EMI and ESD characterization tests

EMI characterization tests should be done for the range of requirements shown in Figure 7 according to MIL-STD-461 and MIL-STD-462 or equivalent. Generally, EMI tests are not required at the component level, unless the considered part is expected to emit a strong EMI signature (e.g., phased emitters), or, alternatively, if it is extremely susceptible to EMI (e.g., a detector component).

Three different models exist for ESD characterization – human body model (HBM), charge-device model (CDM), and machine model (MM). These tests are standard for the electronic industry, and are used to categorize electronic components with regards to the ESD voltage surge, which the part can withstand, regardless of polarity. The three models simulate different scenarios in terms of load, capacitance, and pulse shape, which encompass most known causes for ESD and EOS conditions. HBM generates the expected part performance during handling, which is the leading cause for ESD and EOS damage to electronic devices. The CDM susceptibility is also important for test, qualification, and assembly processes.

The assessment of ESD sensitivity of a spacecraft in operation (e.g., in the Earth plasma environment) is a fast evolving area due to the wealth of recently acquired data. However, similarities with the HBM exist. The spacecraft capacitance is of the order of 100 pF, similar to that of a human body, and the pulse shape closely resembles that used by the HBM.

Class	Voltage Range
0	<250
1A	250 – 500
1B	500 – 1000
1C	1000 – 2000
2	2000 – 4000
3A	4000 – 8000
3B	≥ 8000

HBM ESD testing should be done as described by MIL-STD-1541A, or equivalent. Ideally, all electronic devices should be tested for sensitivity to ESD. The standard categories are shown in the table (left).

During assembly, flight hardware is usually treated as category 1A, unless the ESD sensitivity is specified. However, a large number of flight components are class "0". The trend is that devices sensitivity will increase in the future; thus, more ESD protection must be engineered in the design. ESD risks can also be mitigated on a spacecraft level, by using suitable materials and by avoiding exposed insulators.

Recently, internal ESD (IESD) has come into the focus of attention. IESD has been identified as the leading mechanism for generating electrical overstress in many occasions. The conditions for IESD are generated by high-energy electrons (~MeV), which penetrate the shielding and stop in the insulating package of an electronic device. Continuous exposure leads to the accumulation of trapped charge in the dielectric. In turn, this creates an electric field with a strong gradient near the surface of the insulator. Upon exceeding the dielectric breakdown strength of the insulator, the electric field generates IESD. Severe IESD damage to electronic devices is explained by the proximity of the active regions to the discharge area.

VI. SENSITIVE PARAMETERS PER DEVICE TYPE

All circuit applications of each device type must be evaluated for the effects of parameter variations due to the combined effects of the environments. All device parameters must be within the specified range before, during and after each test. The order with which the different environmental tests are performed must be determined with considerations of the actual application, so that the probability for detecting defects from preceding tests is maximized.

The analysis is typically performed by linearly adding the maximum parameter variation for each effect and verifying acceptable performance of the circuit. As a minimum, all device parameters critical to circuit operation should be included. Device parameters that are known to be particularly sensitive to environmental effects are listed by device type below. This is by no means a complete list but does provide, in order of significance, a focus on the parameters usually of most importance.

Table IV. An example of sensitive parameters, which are likely to be affected by one or more environmental tests.

Device Type	**Sensitive Parameters**
Bipolar Op-Amps	Input Bias current, input offset voltage, gain, output short circuit current
Bipolar regulators/references	Output voltage, maximum load current
Bipolar digital	Delay times, input thresholds
Bipolar transistors, HBT	Current gain, base leakage current
CMOS linear	Supply current, input offsets
MOS Mux/Switch	Leakage currents, input thresholds, functional failure
MOS transistors	Gate threshold voltage, drain to source leakage current
CMOS digital	Supply current, input threshold voltage, functional failure, delay times
DRAM / SRAM	Bit error
CCD's	Dark current
RF HEMT	Frequency shifts, drain-source leakage current, transconductance
Optocouplers	Current transfer ratio
Fiber optics	Transmissivity
LED	Light output
Laser diode	Light output (efficiency), wavelength shift, threshold current
Solar cells	Fraction of maximum power remaining
Detectors	Spectral response, sensitivity, signal-to-noise

VII. EXAMPLES FOR USING THE GUIDELINES

The following example demonstrates the use of the guidelines in this document. An active pixel sensor is considered for use in LEO, GEO, and Europa orbiters, in deep space exploration, and in a Mars Rover.

1. Active pixel sensor (APS)

The active pixel sensor uses CMOS technology with architecture resembling that of DRAM. Although the use of the APS differs greatly from that of a DRAM, the device will perform similarly in the described environments. The expected APS performance is given in *Table V*.

- Radiation environment

  Total dose is not expected to be an issue for APS use in LEO. The performance of an APS in a 1 year GEO mission and in long lasting deep space missions is undefined by the technology itself. Device design and the maturity of manufacturers technology are important factors. The use on a Mars rover requires characterization. The APS cannot be used on long-duration GEO and Europa missions without special measures. These can be using radiation-hard design, or sufficient shielding, or, most likely, both. Displacement damage is not expected to be an issue for any of the considered missions. LET characterization is required; for GEO and long DS missions the performance may or may not be satisfactory.
- Thermal environment

  The use of APS in a protected thermal environment raises no issues, except on a Mars rover, for which characterization is required. Characterization is required for external use in all of these missions.
- Vibration environment – major effects are unlikely to occur.
- EMI / ESD environment

  EMI environment is irrelevant to the APS use. The device, however, may be susceptible to ESD. Assessment is required.

Table V. Expected APS performance in various missions.

major effects unlikely assessment needed unknown performance protection needed		LEO		GEO		Mars R	DS	Europa O.
		1 yr	3 yrs	1 yr	10 yrs	1 yr	10 yrs	1 yr
Radiation	TID							
	Displ.							
	LET							
Thermal	Internal							
	External							
Vibration								
EMI / ESD	EMI							
	ESD							

ACKNOWLEDGMENTS

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APPENDIX

APPENDIX

Radiation data and remarks used to generate *Table I*

When comparing the performance of electronic technologies in radiation environments, it should be noted that the order of importance of the environment parameters varies. The ranking of this importance for electronic and opto-electronic components is the following (ELDRS is a special case of TID and is not listed separately):

- Electronic components: SEE, TID, DD
- Opto-electronic components: DD, SEE, TID

Considering as an example a mission in the Jovian environment, a comparison of TID hardness of a CMOS device and an LED is inadequate. The CMOS will fail due to total dose, whereas the LED fill fail due to displacement damage. The following value can be used as a reference level for consideration of DD in opto-electronic components:

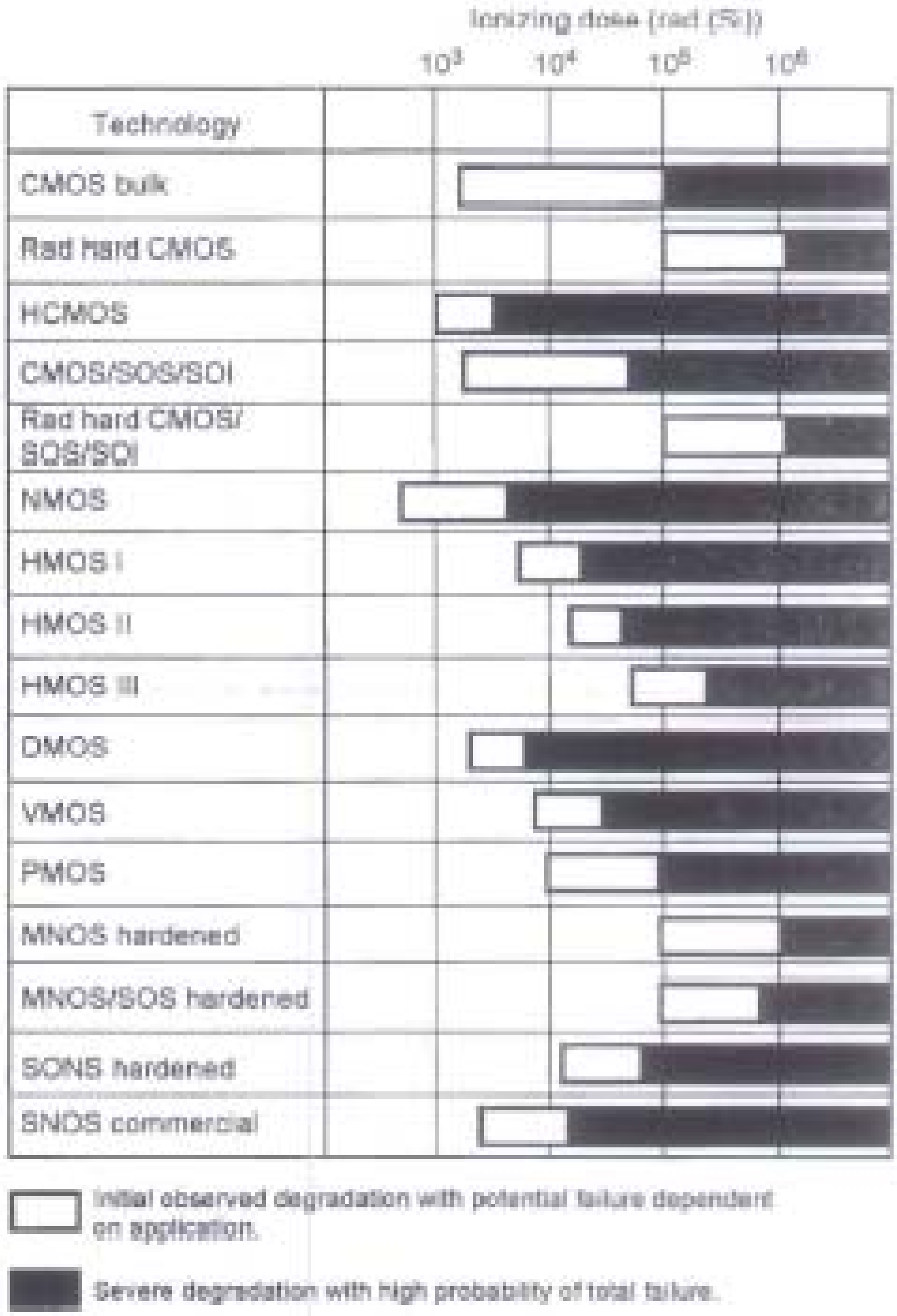
- 1×10^{10} (50 MeV) protons/cm^2

The data used to comprise Table I (following tables) are taken from the following reference:

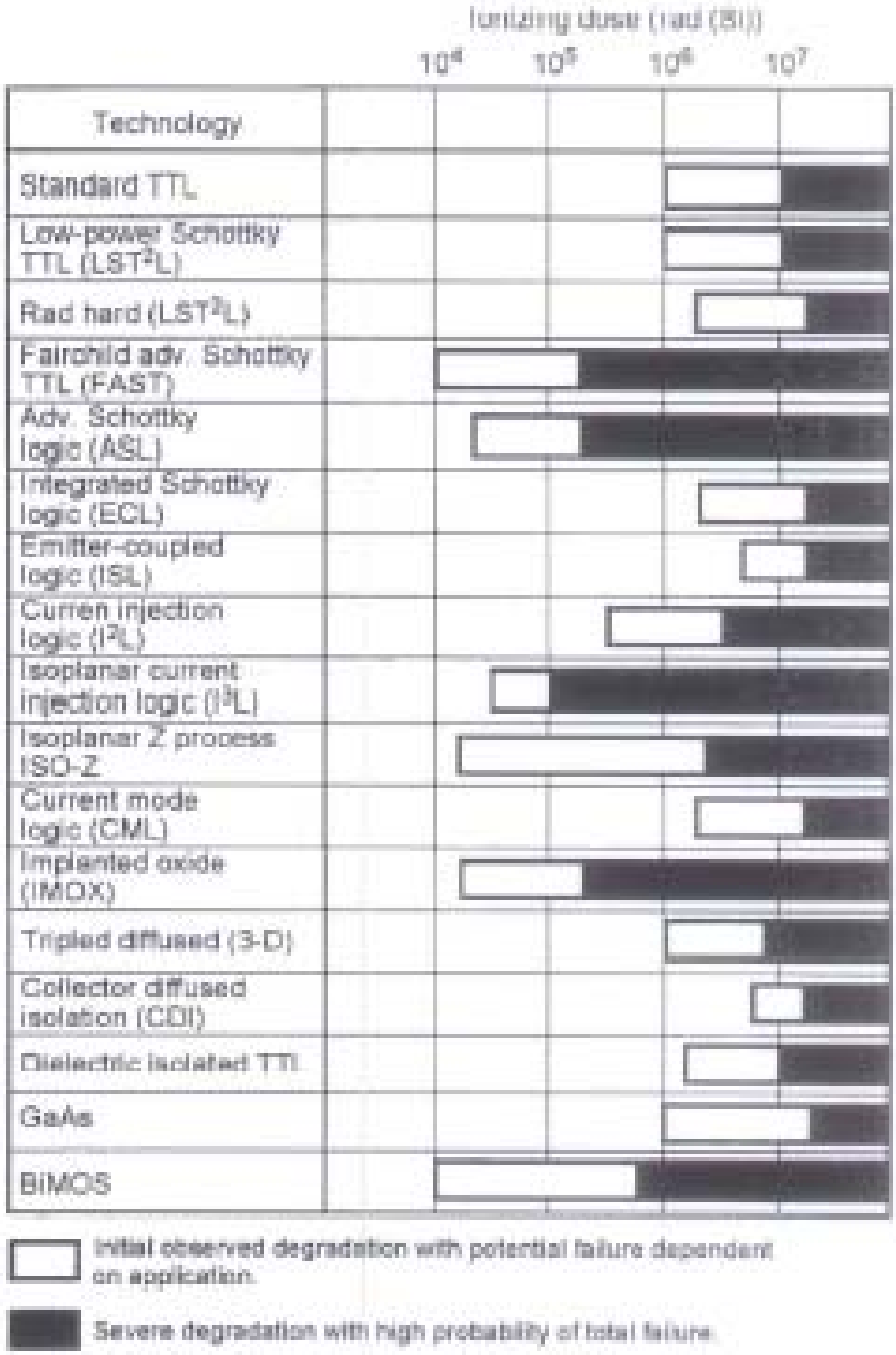
- M. Rose, “updated Bar Charts of Device Radiation Thresholds”, Physirton Corp., San Diego, CA, 1990.

More information can be obtained through the Radiation and FA web site:
http://parts.jpl.nasa.gov/

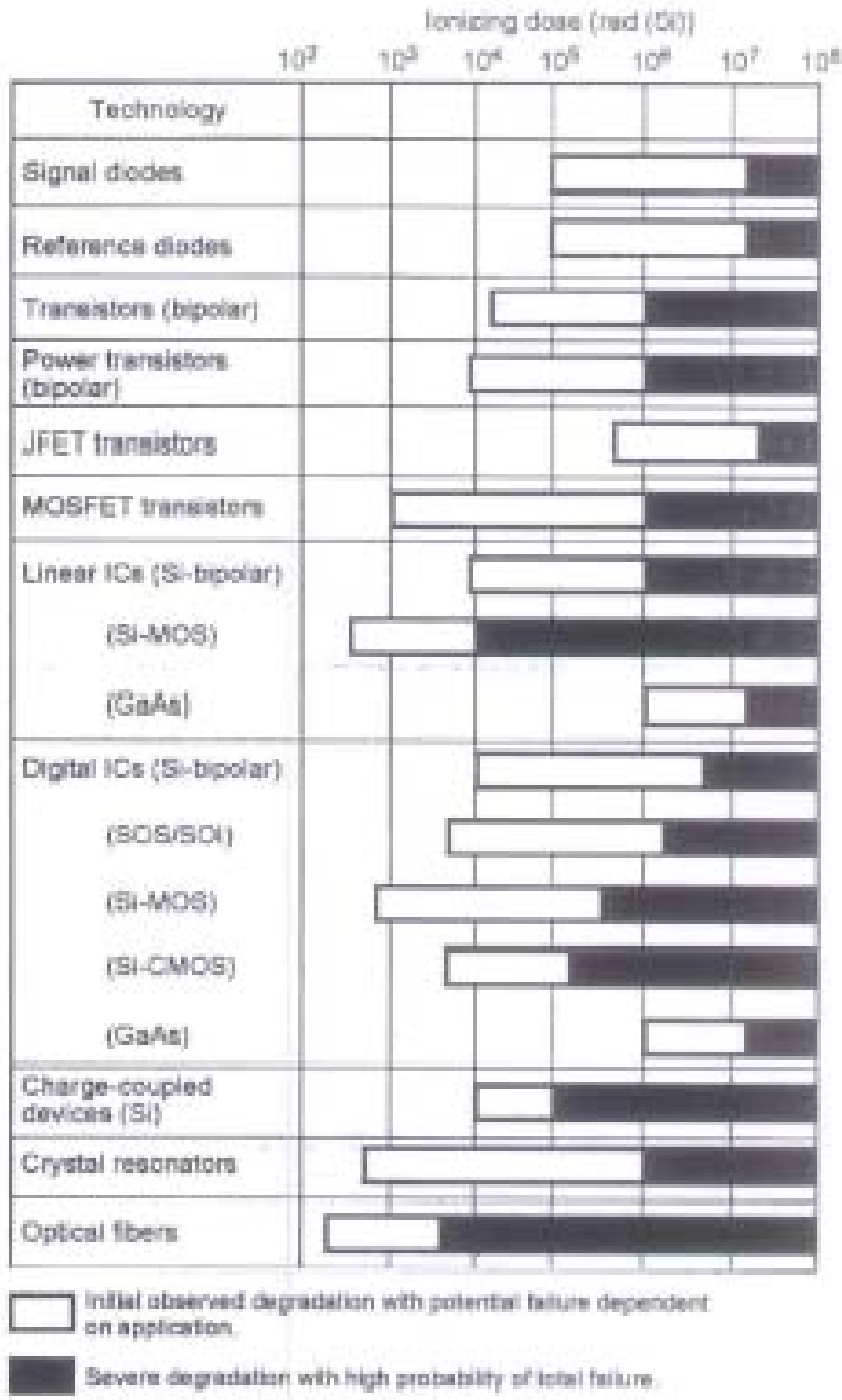
Ionizing dose failure levels for MOSFET integrated circuits



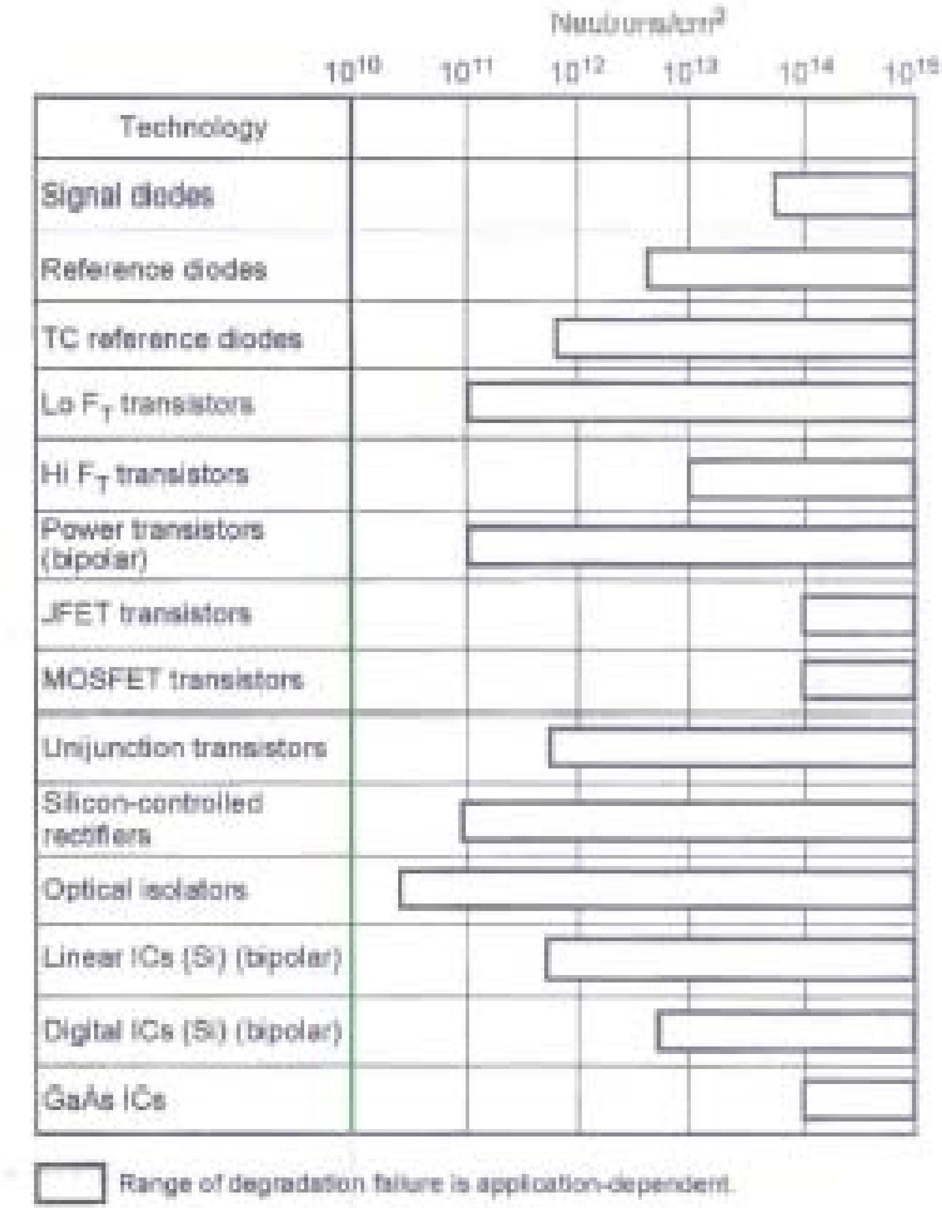
Ionizing dose failure levels for bipolar integrated circuits



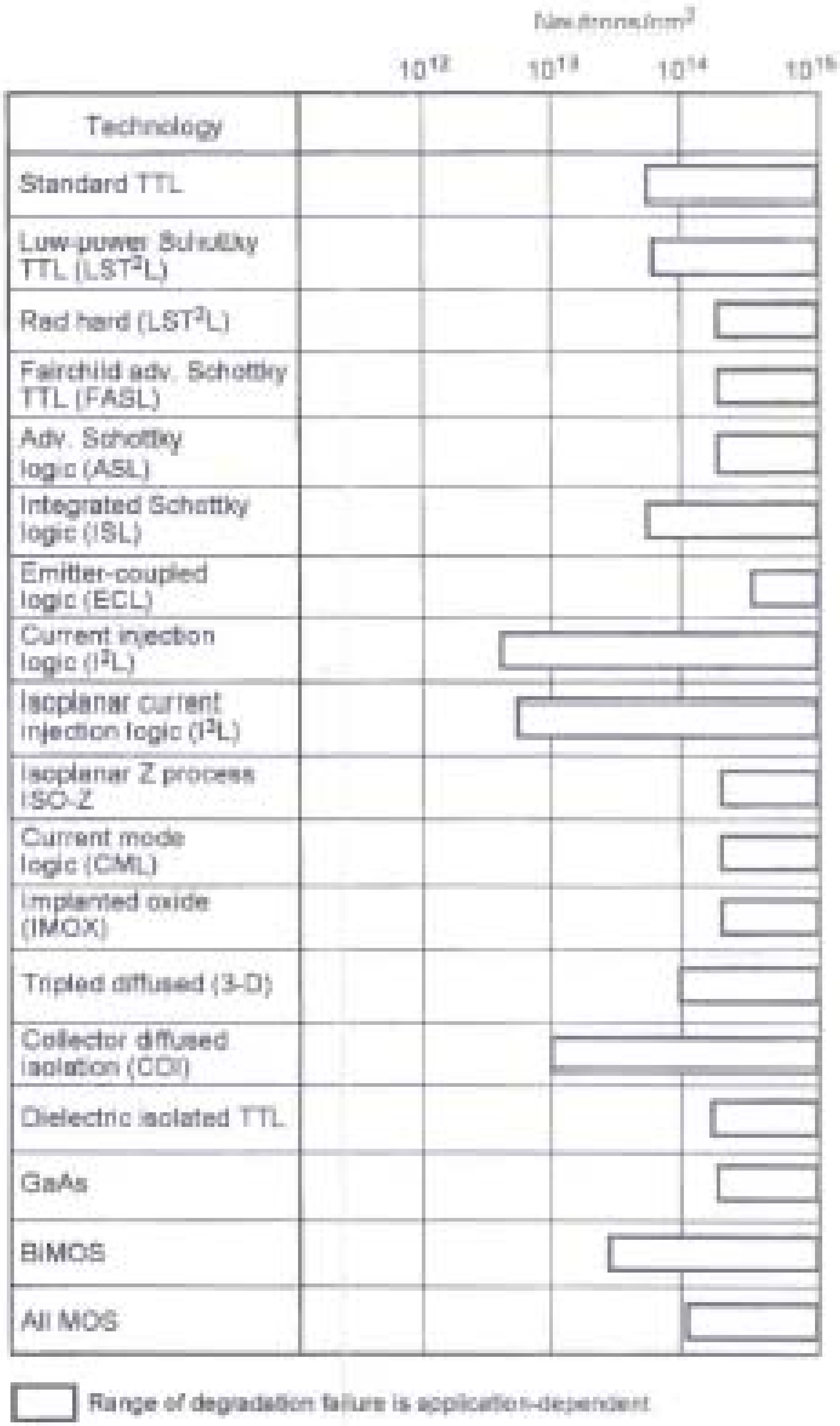
Ionizing dose failure levels for discrete, linear, and digital device families



Neutron hardness levels for discrete devices



Neutron hardness levels for integrated circuit families



The Effects of Space Environments on Electronic Components

Appendix: References

February 2, 2003

This appendix directs the reader to valuable references used to generate the guidelines for the performance of electronic components in space environments, requested by the NMP Office. Enclosed are a CD with the references listed on the next page, and a hard copies of documents obtained from various sources.

References on space environments:

1. D. Hastings and H.B. Garrett, “Spacecraft-environment interactions”, Atmospheric and Space Science Series, ed. A.J. Dessier, Cambridge Univ. Press, 1996.
2. R.N. DeWitt, D.P. Duston, and A.K. Hyder, eds., “The behavior of systems in space environments”, Kluwer Academic Publishers, 1994.
3. J. Feynman, “Solar wind”, Ch. 3: “Handbook of Geophysics and space environment”, Springfield, VA: National Technical Information Service, ADA16700, 1985.

Note: A course entitled “Spacecraft-Environment Interaction” is given periodically at JPL by Dr. H.B. Garrett, Section 513.

References on radiation effects on electronic components:

1. N.J. Rudie, “Principles and techniques of radiation hardening”, Vols. 3-7, 3rd ed., Western Periodicals, 1986.
2. G.C. Messenger and M.S. Ash, “The effects of radiation on electronic systems”, 2nd ed., Van Nostrand Reinhold, 1992.
3. A. Holmes-Siedle and L. Adams, “Handbook of radiation effects”, 2nd ed., Oxford University Press, 2001.
4. K.A. LaBel et al., “A compendium of recent optocoupler radiation test data”, 2000 IEEE Radiation Effects Data Workshop Record.

Note: A course entitled “The Effects of Space Radiation on Microelectronics” is given periodically at JPL by the Radiation Effects Group, Section 514.
Presentation available on-line at http://parts.jpl.nasa.gov/docs/Radcrs_Final.pdf

References on spacecraft charging:

1. H.C. Koons et al., “Theimpact of the space environment on space systems”, 6th Spacecraft Charging Technology Conference, AFRL-VS-TR-20001578, 1 September 2000.
2. A.R. Frederickson, “Upsets related to spacecraft charging”, IEEE Trans. on Nucl. Sci., vol. 43, No. 2, April 1996.
3. A.R. Frederickson, “New scaling laws for spacecraft discharge pulses”, 7th Spacecraft Charging Technology Conference, ESTEC, Noordwijk, The Netherlands, November 2001.

References enclosed on the CD:

Standards:

1. JPL document D-17868: “Design, verification/validation and operations principles for flight systems”
2. MIL-HDBK-338B: “Electronic reliability design handbook”
3. MIL-HDBK-340: “Application guidelines for MIL-STD-1540B; Test requirements for space vehicles”
4. ESA PSS-01-702: “A thermal vacuum test for the screening of space materials”
5. ESA PSS-01-704: “A thermal cycling test for the screening of space materials and processes”

Data references:

1. K.L. Bedingfield, R.D. Leach, and M.B. Alexander, Editor “Spacecraft system failures and anomalies attributed to the natural space environment”, NASA Reference Publication 1390, August 1996.
2. J. Newell and K. Man: “Mission specific environmental testing”
3. Emilio Beltran: “Thermal (…analysis of Dawgstar in LEO Orbit)”
4. Dan Butler: “Carbon-Carbon Radiator” (on EO-1)
5. Aerospace and Ocean Engineering Department, Virginia Tech, Blacksburg, VA: “Structures, mechanisms, launch vehicle selection”
6. Sam DiMaggio, The Aerospace Corporation, “Structural design and verification of space and launch vehicles”
7. Rosa Leon: “Advanced III-V devices for potential NASA applications”
8. Insoo Jun: Calculation of radiation in Earth orbits (LEO, PEO, MEO, GEO) in 4π geometry.

Internet links:

1. Earth's Thermal Environment
   http://www.tak2000.com/data/planets/earth.htm
2. Radiation and the International Space Station: Recommendations to reduce risk:
   http://www.nas.edu/ssb/radissmenu.htm
3. Army space reference text:
   http://www.fas.org/spp/military/docops/army/ref_text/index.html#CH6
4. Catalogue of ESA procedures standards and specifications:
   http://esapub.esrin.esa.it/pss/pss-cat1.htm